AD-A061 792 ROYAL AIRCRAFT ESTABLISHMENT FARNBOROUGH (ENGLAND) F/G 21/3
THE USE OF ION THRUSTERS FOR ORBIT RAISING.(U)
JUN 78 D G FEARN
UNCLASSIFIED RAE-TR-78068 DRIC-BR-64931 NL

| OF |

AD
A061792

END
DATE
FILMED
2 79
DDC

TR 78068

ROYAL AIRCRAFT ESTABLISHMENT

*

Technical Report 78068

June 1978

THE USE OF ION THRUSTERS FOR ORBIT RAISING

by

D.G. Fearn

*

Procurement Executive, Ministry of Defence
Farnborough, Hants

18 DRIC 19 BR-64931

UDC 621.455.4 : 629.19.014.9 : 629.19.085

12 33p.

ROYAL AIRCRAFT ESTABLISHMENT

9 Technical Report 78068

Received for printing 27 June 1978 11

6 THE USE OF ION THRUSTERS FOR ORBIT RAISING

by

10 D. G. Fearn

14 RAE-TR-78068

SUMMARY

Earlier analytical results for changing the orbital altitude of a spacecraft, using a tangentially thrusting electric propulsion system, have been employed to confirm that the mass of propellant required to attain a geostationary orbit is only a very small fraction of the initial mass of the satellite. However, such an orbit transfer technique requires a relatively long period of time compared with chemical propulsion, typically 100 to 300 days, and, for spacecraft masses of 1000 kg or more, multi-kW solar arrays are necessary. Unfortunately, the long transfer time also leads to a significant degradation of the power produced by these solar arrays, owing to the impact on them of energetic particles while traversing the earth's radiation belts. Nevertheless, it is concluded that this technique for manoeuvring a satellite into synchronous orbit can offer attractive economic benefits for two broad classes of spacecraft. These are relatively small satellites, of 1000 kg or less, for which a dedicated ion thruster system can be advocated, and much larger devices, for which a reusable solar electric tug vehicle having its own solar arrays would be appropriate. In the former case, a payload ratio of about 0.7 could be achieved, with a transfer time of 150 days, and two or three spacecraft could be launched on a single Ariane vehicle.

Departmental Reference: Space 551

LIST OF CONTENTS

1 INTRODUCTION

Most spacecraft are fitted with an on-board propulsion system which is used to either change or maintain the satellite's orbital parameters and attitude. In operation, such a propulsion system changes the total energy and momentum of the spacecraft by the expulsion of a stream of exhaust gases at a high velocity. Each different manoeuvre requires that the spacecraft be given a particular velocity increment, which is equivalent to a specific momentum change. Equating this momentum to that in the exhaust of the propulsion system indicates that the mass of propellant used is inversely proportional to the exhaust velocity achieved. If a large velocity increment is called for, the propellant required can be a significant fraction of the initial mass of the spacecraft.

The exhaust velocity of conventional thruster systems, either bi-propellant or mono-propellant, is limited by the energy available in the associated chemical reactions. Even the addition of an electrical heating stage to the exhaust, as in the power augmented electrothermal hydrazine thruster (PAEHT)[1], does not increase the exhaust velocity much above 3 km/s, and the technology involved is very demanding for long duration missions. Of gas jet systems, only the hydrogen resistojet[2], with an exhaust velocity of up to about 8 km/s, offers a significant advantage.

To achieve an order of magnitude reduction of propellant mass, as compared with a conventional hydrazine thruster system, exhaust velocities of 30 km/s or greater must be attained. This can be done by a variety of electric propulsion (EP) devices, in which the propellant is electrically charged or ionised so that it can be accelerated by electrostatic or electromagnetic forces. The type of device which is at present most favoured for this application is the Kaufman ion thruster, which is under development in the USA, the UK, France and Japan. A similar concept being studied in Germany employs an RF discharge rather than the dc discharge in the Kaufman thruster.

The use of ion thrusters for north-south station-keeping (NSSK) has been widely advocated[3,4], and it has been shown that the mass saving for a typical long-life synchronous communications spacecraft would be very substantial. This would provide important economic benefits, through either the attainment of greater reliability, the use of smaller launch vehicles, or the provision of increased communications capacity.

The mass advantage of using EP systems becomes even greater for missions requiring large velocity increments, such as major changes of orbital altitude or
068 inclination, and interplanetary probes. In the case of altitude changes, a mission

of major interest is the transfer of applications spacecraft from low parking or transfer orbits to synchronous orbits. This normally involves the use of large quantities of propellant in perigee and apogee boost motors, and any reduction in this requirement would considerably enhance payload capability. As an example, in the Intelsat V spacecraft[5], 920 kg of the initial 1870 kg in transfer orbit is devoted to the apogee motor and its propellant. As will be shown later, the use of ion thrusters might allow this propellant mass to be reduced by perhaps 60 to 70%.

The use of EP in this context has been studied for many years in the UK and the USA. The RAE was involved as early as 1963, when King-Hele[6] suggested an orbit expansion manoeuvre employing tangential thrusting. The concept was generalised to many different missions by Burt[7,8] and, simultaneously, some very detailed analysis was undertaken for ELDO by Davidson[9] and Sarnecki[10]. These studies all confirmed the feasibility of using a slow orbital expansion technique to reach the desired orbit with the minimum expenditure of propellant, as did work done independently in the USA by NASA[11] and Comsat[12]. A further extension by Ives[13] took into account the degradation of the solar arrays powering the ion thrusters during the manoeuvre, and the Rocket Propulsion Establishment (RPE) included the interesting case of using a dual thruster system to optimise the effective exhaust velocity[14].

At the time of the earlier RAE studies, suggestions were made for commencing a research and development programme aimed at producing an ion thruster specifically for the orbit raising mission. In 1967 it was decided to follow this course of action, a thrust of 15 mN being selected as appropriate to the requirements of the small spacecraft intended for launch on the Black Arrow rocket. During the development of the 15mN T2 thruster[15], a spacecraft intended specifically for the spiral orbit-raising mission was designed[16]. However, with the demise of the Black Arrow launcher and of UK interest in ELDO, it was decided to concentrate development on the NSSK mission. Consequently, the later T4A and T5 thrusters[17] were designed for that application, with a lower nominal thrust of 10 mN.

More recently, the possibility of launching spacecraft into low parking orbits by means of the NASA Space Shuttle has created a demand for the development of a 'tug' vehicle which would be able to manoeuvre spacecraft into their final orbits. Although chemically-propelled tugs have been suggested, particularly as an interim measure, the use of EP is very attractive. A modular solar electrically-propelled tug has been studied very extensively by NASA[18], and the University of Southampton[19] and the RPE[20] have also examined this concept in detail, with emphasis in all cases 068

on reusable systems. However, to give acceptable transfer times for large spacecraft, the tugs under consideration have invariably included large thrusters and solar arrays, typically six or eight 30cm thrusters each consuming 2 to 4 kW.

It will be shown below that present spacecraft designed for communications and similar missions can be positioned in synchronous orbit using EP, but with rather less propellant than has been assumed in the studies of possible solar electric tugs. Transfer times are acceptable, provided that appropriate satellite replenishment strategies are adopted, and the thruster technology required is a reasonable extrapolation of that represented by the T4A and T5 devices[17]. Although most emphasis is placed on payloads launched by the ESA Ariane vehicle, the results are also applicable, with some modification, to the Space Shuttle. In the latter case, the design and construction of a specialised reusable tug smaller than that advocated by NASA might be justified.

2 THEORY

If all significant effects are to be taken into account, including the degradation of solar arrays as the spacecraft passes through the radiation belts around the earth, atmospheric drag at the perigee of the transfer orbit, and the shadowing of solar arrays during eclipse, the analysis becomes complex. However, Davison[9] and Sarnecki[10] have shown that the second and third of these influences have only a small effect on the mass transferred to the final orbit and that a simple correction factor can be applied to give, with reasonable accuracy, the transfer time.

The extension of the theory to take into account solar array degradation, together with the removal of eccentricity and an inclination change, was undertaken by RAE for ELDO[13] and adds very considerably to the magnitude of the computations involved. Although the computer program devised for that study could have been adapted for use in the present investigation, this did not seem worthwhile in view of the tentative nature of the mission and the availability of results from the earlier work which are of general application. In particular, a single curve was presented[13] as a guide to the power loss suffered during the transfer to a synchronous orbit as a function of time taken. This curve represented all 5° inclined parking orbits with apogee heights within the range 500 to 5000 km, and all exhaust velocities between 20 and 60 km/s. Both circular and elliptical parking orbits were included, and the results are applicable, to a first approximation, to the present case.

Following Sarnecki[10], if the effective exhaust velocity of the thruster system is v_e, the initial mass of the spacecraft in an equatorial parking orbit is M_o and the mass at time t after the manoeuvre has commenced is M_t, then

$$\frac{M_t}{M_o} = \exp\left[\frac{\mu^{\frac{1}{2}}}{v_e}\left(\frac{1}{a_t^{\frac{1}{2}}} - \frac{1}{a_o^{\frac{1}{2}}}\right)\right] \quad , \tag{1}$$

where a_o and a_t are the semi-major axes of the initial orbit and of the orbit at time t, and μ is the gravitational constant of the earth (3.986×10^{14} m^3/s^2). For a synchronous final orbit, $a_t = a_f = 42164$ km and M_t becomes M_f. The above equation applies strictly to the drag and eclipse free case, but it also represents an excellent approximation when these two factors are present[9].

Equation (1) is applicable to both circular and elliptical parking orbits[10] and relates the mass of the satellite at any time during the manoeuvre to the initial and instantaneous magnitudes of the semi-major axis. However, Davison[9] concluded that the difference in transfer time and propellant mass between manoeuvring from a circular parking orbit and from elliptical parking orbits having the same energy level (*ie* the same semi-major axis) is negligible. In addition, the initial orientation of an elliptical parking orbit also has a negligible effect. Consequently, it has been assumed in the present study that circular parking orbits are employed, but the results presented are also relevant to elliptical orbits, provided that their perigees are not so low as to cause a significant increase in the average atmospheric drag.

In order to make use of equation (1), the effective exhaust velocity must be related to measured thruster characteristics. The velocity v of the beam ions can be deduced from the net accelerating voltage V_T. Thus

$$v = \left(2e\,\frac{V_T}{m_i}\right)^{\frac{1}{2}} \tag{2}$$

for singly-charged ions, where e and m_i are the ionic charge and mass respectively. However, a certain proportion of the propellant used by a thruster is not ejected in the beam at this velocity, but is lost as neutral vapour or as low energy ions. This causes v_e to be less than v by a significant factor. 068
This factor can be deduced by considering the thrust F_T produced by a thruster

with a total propellant consumption rate $\dot{m}_T$ and an overall mass utilisation efficiency η_m . Equating thrust to rate of change of momentum,

$$F_T = \dot{m}_T v_e = \eta_m \dot{m}_T v \ ,$$

ie

$$v_e = \eta_m v \ . \tag{3}$$

The time T taken to execute a given orbit transfer using total thrust F is, in the absence of shadowing or air drag[9], given by the equation

$$FT = M_p v_e \ .$$

In the presence of drag and shadowing, the time is increased to T_s by a 'shadowing factor' $\bar{U}$, so that

$$FT_s = M_p \frac{v_e}{\bar{U}} \ . \tag{4}$$

Here, $M_p = M_o - M_f$ is the mass of propellant consumed during the transfer. The factor $\bar{U}$ cannot be derived analytically, but values have been tabulated by Davison[9] and Sarnecki[10] for exhaust velocities of between 8 and 50 km/s and transfers to geostationary altitudes from circular orbits with initial heights h_o of 350, 500 and 650 km. Over this wide range of conditions, $\bar{U}$ varied from 0.825 to 0.865, which was less than the effect due to the time of launch. It was therefore suggested[9] that it would be valid for most purposes to adopt a mean value, but in the results presented here extrapolated values have been employed.

The power needed for a given transfer manoeuvre is of great importance, because it will determine the size, weight and cost of the solar arrays required. Conversely, if the power available is a mission constraint, this will determine the thrust and exhaust velocity and therefore the transfer time.

The power P_T consumed by a thruster giving thrust F_T is the sum of the beam power, the discharge power, and the power used by auxiliary components. If the latter contribution is P_a , it can be shown[21] that

068

$$P_T = F_T\left(\frac{v}{2} + \frac{\varepsilon}{v}\right) + P_a \ , \tag{5}$$

where ε is the discharge chamber efficiency, measured in eV/ion multiplied by the ionic charge to mass ratio. Due to losses in the power conditioner supplying the thruster, which has an efficiency η_P , the input power to a single ion thruster system is, from equation (5),

$$P_i = \frac{1}{\eta_P}\left[F_T\left(\frac{v}{2} + \frac{\varepsilon}{v}\right) + P_a\right] . \tag{6}$$

In calculating the total mission power P required to give thrust F with beam velocity v , it is first necessary to decide upon the number n of thrusters to be used, since the proportion of P due to auxiliary power requirements is dependent on this. Once n is known, $F_T = F/n$ and $P = nP_i$. Thus F_T can be determined from equation (4), for a given mission time, and P from equation (6).

3 RESULTS

3.1 Mission and thruster constraints

The calculations reported below are based on the use of the Ariane launcher. This vehicle has a design maximum payload[22] of 2500 kg, which can be placed into a 1000km altitude circular orbit, or an elliptical orbit with an apogee height of up to 10000 km and a perigee height of 200 km (Fig 1). Of course, smaller payloads can be launched into higher circular orbits or into elliptical orbits with greater apogee heights. However, as it was not possible to cover all possibilities, emphasis has been placed on the 1000km circular orbit. In all cases, a geostationary final orbit has been considered, because this is of the greatest commercial interest.

It has been assumed that a typical orbit transfer would be accomplished by using an array of thrusters operating simultaneously. As well as providing redundancy, this would enable relatively small thrusters to be employed, thus avoiding the necessity of developing very large devices specifically for this mission. In addition, as mentioned in sections 3.4 and 3.6, it would also be possible to match the thrust to the power available[13], thereby reducing the overall mission time.

Total thrust levels of 50 to 1000 mN have been considered. The lower values, up to about 100 mN, could be achieved by using multiple T4A or T5 thrusters, uprated to around 20 mN by methods already well established. The intermediate values would probably require the development of a new thruster, 068

perhaps of 15 cm diameter. At the highest thrust levels, a 100mN device would be desirable, which would be about 25 cm in diameter. For any particular mission, the aim would be to use not less than three thrusters and not more than ten, thereby retaining a balance between flexibility and complexity.

The electrical efficiency of an ion thruster is a strong function of v. It is therefore preferable to use as high a value of v as possible, within the limitation imposed by any power restrictions. The velocity range considered here is 25 to 70 km/s. As shown in Fig 2, this corresponds to an accelerating potential, for singly-charged mercury ions, of 0.65 to 5 kV, these values having been derived from equation (2). Also indicated in Fig 2 are values of v_e appropriate to $\eta_m = 0.88$, which is a typical utilisation efficiency for a thruster of 10 to 30 cm diameter[3].

3.2 The mass in geostationary orbit

Equation (1) has been used to derive the ratio M_f/M_o as a function of v_e (or v) for initial circular orbits at altitudes of between 300 and 5000 km. The results, presented in Fig 3, show that mass ratios in excess of 90% are readily available for high exhaust velocities. Thus, from the point of view of payload, EP is evidently an attractive method of manoeuvring spacecraft into geostationary orbit, provided that adequate time and power can be made available.

For altitudes below 2000 km, the initial orbit has a relatively small influence on the mass ratio. However, a large gain is made if h_o is increased to 5000 km or more. It may therefore be concluded that an Ariane launch into a 1000km circular orbit gives only a marginal improvement in mass ratio, compared with a Shuttle launch into a much lower orbit (350 km). For example, at $v = 40$ km/s, the Shuttle would give $M_f/M_o = 0.876$, whereas Ariane would increase this to 0.886, an improvement of only 1%.

The mass ratio can be increased slightly if an elliptical parking orbit is used. For example, assuming that the maximum Ariane payload is required, Fig 1 shows that an apogee height of around 9000 km is attainable with a perigee of 300 km, giving, for a circular orbit of the same energy level, $h_o = 4650$ km. From Fig 3, M_f/M_o is then about 0.92 for $v = 40$ km/s.

The mass in synchronous orbit can be calculated from the data in Fig 3 for any set of initial conditions. As mentioned in section 3.5, however, this will not represent the useful payload, because the ion thrusters and at least part of the power source will not be needed after completion of the orbit transfer.

3.3 The transfer time

Substituting $M_p = M_o - M_f$ in equation (4),

$$FT_s = \frac{M_o v_e}{\overline{U}} \left(1 - \frac{M_f}{M_o}\right) , \qquad (7)$$

which gives an excellent guide to the time required for the orbital transfer at a given thrust level. As already mentioned, to make use of this expression, values of $\overline{U}$ were employed (Fig 4) which were extrapolated from those given in Refs 9 and 10. This extrapolation was for orbital altitudes of up to 2000 km; although the greatest errors in the present analysis were associated with this process, particularly beyond 1000 km, they did not exceed $\pm 1.5\%$.

Values of FT_s are plotted against v_e and v in Fig 5 for initial circular orbits at altitudes of 300, 1000 and 2000 km and for M_o = 1000, 1500, 2000 and 2500 kg . It can be seen that the transfer times are quite substantial for reasonable values of F . For example, on the largest spacecraft, F = 1 N would give T_s = 117 to 160 days , depending on the exhaust velocity and initial orbit. For a 1000kg spacecraft, F = 250 mN might be more appropriate, giving times of between 188 and 256 days. To achieve shorter transfer times, the thrust, and therefore the power, would have to be increased.

The results shown in Fig 5 are not strongly dependent on v_e , FT_s varying by less than 10% over the range of v_e considered. It will be noted that FT_s , and therefore T_s at constant F , decreases as v_e falls. Consequently, if shorter times are required, lower values of v_e should be used, although, as shown in Fig 3, the penalty will be a reduction of M_f/M_o , which becomes increasingly severe for exhaust velocities of below 30 km/s.

An alternative method of reducing FT_s is to increase the semi-major axis of the initial orbit. As shown in Fig 5, this has a much larger effect than altering v_e , and also gives a higher mass ratio.

The influence of the selected thrust on T_s is shown in Fig 6, for a transfer from a 1000km circular orbit. It will be seen that, as described above, v_e has only a marginal effect.

3.4 Power requirements

The total power P required was calculated as outlined in section 2. This calculation assumed the availability of the thrusters detailed in Table 1 below. 06

Table 1

ASSUMED THRUSTER CHARACTERISTICS

Diameter (cm)	Maximum beam current (A)	Discharge power (W)*	P_a (W)	Thruster mass (kg)	Gimbal system mass (kg)
10	0.27	44	24	1.5	1.0
15	0.6	128	37	3.0	3.0
25	1.7	350	61	6.0	3.0

* includes keeper discharge

In deriving this table, the values for the 10cm thruster were based on the T5 device[17,23], those for the 15cm thruster on SERT II[24], and those for the 25cm thruster on the NASA/Hughes 30cm device[25].

The results obtained, when plotted as curves of P against T_s, do not vary smoothly because of the need to change the number of thrusters used as F is altered. Only when F is such that all the installed thrusters can operate at full efficiency (*ie* at maximum beam current) are the best results obtained. At intermediate values, when maximum efficiency is not being attained, P is slightly higher than would be the case if a thruster were available to give peak efficiency at the particular values of n and F in question. The effect is largest with the smaller thrusters operating at low v, when only a few are being used, and even then it is small. For example, in the case of a 1000kg spacecraft which requires slightly more thrust than can be provided by four 15cm thrusters, the power penalty is 165 W (discharge plus auxiliary power) in a total of, say, 3.3 kW, if $v_e = 25$ km/s. This penalty is only 5%, so, in view of the other approximations made in the calculations of P, its effect has been ignored.

Results are shown in Fig 7 for a 2500kg spacecraft with initial circular orbits of 300 and 1000 km altitude and $v_e = 25, 30, 40$ and 50 km/s. It can be seen that transfer times are long, that power levels are very high, and that raising v_e at a given value of P causes a large increase in T_s. The altitude of the initial orbit has a relatively minor influence on the power required, about 2 to 4 kW at the lower values of v_e.

An inescapable conclusion is that, for reasonable transfer times, the power levels are such that it is most unlikely that the spacecraft would be carrying sufficiently large solar arrays. Consequently, if such arrays were provided for the orbital transfer, they would probably not be used during the operational life of the spacecraft, making this an economically unsound proposition. This
)68 conclusion is even more certain, if the additional array area needed to take

account of degradation during the orbit expansion[13] is included. However, if the requirement that the thrusters and power source be dedicated to the spacecraft in question is removed, these then form a separate tug vehicle[19,26], which can be reused[20], and which can be fully justified as part of an integrated space transportation system.

The results for a 1000kg spacecraft with an initial circular orbit of 1000km altitude are shown in Fig 8. This corresponds more nearly to future communications satellites, which might be provided with multi-kW solar arrays. In this case, transfer times of below 200 days are feasible with $P = 6$ to 10 kW and, at the lower exhaust velocities, times of 120 to 150 days are possible. Consequently, with spacecraft of this size, it is more reasonable to consider using a dedicated propulsion system specifically for the orbit-raising manoeuvre. A larger solar array than needed operationally would probably be desirable but, after degradation during the manoeuvre, it would more nearly fit the needs of the communications payload. Thus much of the array mass would be ascribed to the actual useful payload, M_{pl} , and the concept becomes very attractive.

3.5 Payload mass in synchronous orbit

The useful payload mass may be calculated by subtracting from M_o the total mass of the propulsion system, which is the sum of the items listed below.

(i) Propellant mass M_p .

(ii) Propellant tank mass, taken as $0.1M_p$. This includes valves, pipelines, etc.

(iii) Solar array mass. This is found by dividing P by the assumed power per unit mass of the array. The latter has been taken to be the end-of-life value for the RAE lightweight array[27], 44 W/kg.

(iv) Power conditioner mass. As large power levels are involved, this has been deduced by reference to the NASA design of a solar electrically-powered tug[26], which uses 18 to 25 kW. The value of the assumed conversion factor is 14.2 kg/kW; this includes provision for active thermal control, wiring, structure, etc.

(v) Thruster mass. This has been estimated from existing designs and is given in Table 1.

(vi) Gimballing system mass. This has been included, although it may not be needed if attitude control of the spacecraft can be performed by throttling selected thrusters.

Due to the fact that the aggregate mass of the thrusters and pcu s changes in steps as n is varied, M_{pl} is not a smoothly varying function of T_s ; each time another thruster and pcu is added, the total mass increases by a finite amount, which can be as much as 109 kg for the 25cm thruster operating with $v_e = 50$ km/s . At lower v_e , this reduces to 40 kg and, for the 15cm thruster, the increment is of order 17 to 42 kg. For clarity, these perturbations have been smoothed in Figs 8 and 9.

This analysis shows that the dominant factor in determining the mass of the propulsion system is the value of P . If high power is needed, the power conditioner and solar array together account for most of the mass, and this factor is usually more important than the trend of increasing M_p at low v_e . Thus, although Fig 3 would suggest that a high value of v_e is necessary to place a large proportion of the original mass into the final orbit, so much of this final mass is taken up by thrusters, power conditioners and solar arrays that it is not worthwhile increasing v_e above about 30 km/s.

The results for the 1000kg spacecraft shown in Fig 8 clearly demonstrate the above points. At high values of T_s and P there is a cross-over of the curves, with $v_e = 30$ km/s being marginally better than 25 km/s, but the calculation errors are of the order of the differences between the curves. More significantly, at about 150 days, which can be achieved with $P = 8$ kW , the ratio M_{pl}/M_o is rather less than 0.5, indicating that longer times would be necessary to make the final payload more attractive. However, if the solar array was considered as part of the useful payload, the situation would be much more favourable, as indicated by the upper lines in Fig 8. In this case, the higher values of v_e have a definite advantage and M_{pl}/M_o can be increased to 0.7, with T_s down to 150 days.

Similar results for the 2500kg spacecraft are depicted in Fig 9 for the 1000km initial circular orbit. Although the useful mass placed in synchronous orbit can be very substantial, for reasonably short transfer times it falls below 50% of the initial mass. Longer times can be tolerated if a tug is used to transfer a totally inactive passenger spacecraft; in this case M_{pl}/M_o could approach 70%.

As an indication of the division of the mass of the propulsion system between its various components, typical examples are given below in Table 2, for $M_o = 1000$ and 2500 kg , $v_e = 30$ km/s , and a transfer time of 157 days.

Table 2

SUB-DIVISION OF PROPULSION SYSTEM MASS

M_o (kg)	Solar array mass (kg)	Power conditioner (kg)	M_p (kg)	Tankage (kg)	Thrusters* (kg)	Thruster diameter (cm)
1000	186	116	133	13	48	15
2500	452	282	332	33	63	25

* including gimballing system

3.6 Degradation of solar array during orbit transfer

An unfortunate characteristic of the spiral orbit-raising mission is the degradation in performance of the solar arrays during transfer. This degradation, which occurs when passing through the radiation belts surrounding the earth, has been considered in detail by Ives[13], for the case of thrust matched to available power. The problem was not amenable to an analytical solution, and numerical methods had to be employed; these included the removal of eccentricity and an inclination change of 5^o.

An important conclusion from the study was that the degradation is hardly related to the height of the initial orbit, for altitudes below 5000 km. In fact, it was shown that an initial 5000km circular orbit gives less than 2% improvement in the final output power of the solar array compared with an initial 500km circular orbit, assuming the same transfer time. This is because, in both cases, the spacecraft spends roughly the same fraction of the transfer time in the highest intensity regions where the energetic particle flux exceeds the equivalent of 10^{15} electrons/cm^2/d.

A further conclusion was that a spacecraft takes a constant fraction of T_s to traverse any particular increment of altitude, at constant F. Consequently, if, for example, T_s is doubled, the spacecraft takes twice as long to increase its altitude by a given amount, and the particle flux encountered is also doubled. Thus the total flux during the complete transfer is directly proportional to T_s, which is the dominant factor in determining array degradation.

The power decay profiles obtained can be represented by the example shown in Fig 10, which gives the fractional degradation as a function of the elapsed proportion of T_s, for M_o = 2780 kg and v_e = 40 km/s. It will be seen that the degradation causes the power available in synchronous orbit to fall to between 61 and 67% of the original power, depending on transfer time. Such degradation 068

ratios are, in fact, typical of all the missions considered, as depicted in Fig 11. This illustrates that the percentage power lost during the transfer is only weakly dependent on v_e and the initial orbit and, for most purposes, the loss can be generalised with sufficient accuracy by use of the bold upper curve. This overestimates the loss by a maximum of 2.5%, which is probably less than the errors associated with changing thrust levels, the effect of the time of launch, etc.

4 CONCLUSIONS

Earlier analytical results for changing the orbit of a spacecraft using a tangential thrusting electric propulsion system, have been used to confirm that the propellant mass required to transfer a satellite into a synchronous orbit is only a very small fraction of the initial mass. The mass ratio M_f/M_o is mainly dependent on the thruster exhaust velocity and can readily exceed 90%.

Two very serious penalties were shown to result from this method of attaining high mass ratios. The first is the requirement for very long transfer times, of the order of 100 to 300 days, and the second is the need for multi-kW solar arrays. In the case of a large satellite and a reasonably short transfer time, the latter requirement increases to tens of kW. The mass of the hardware associated with the electric propulsion system is then so large that, although M_f is high, the useful payload in geostationary orbit is greatly reduced, unless parts of the propulsion system can be used later for other purposes. The mass penalty is particularly serious at high exhaust velocities, because of the need to provide increased power. This effect tends to dominate the increases in mass ratio with rising v_e, and the optimum exhaust velocity is generally around 25 to 30 km/s.

These disadvantages of the low-thrust orbit-raising concept are further amplified when the degradation of solar arrays by the impact of high energy particles is included. These can reduce the final power output by 30 to 40%, depending on the mission duration, and this will considerably increase the mass and cost of the solar arrays.

Nevertheless, it has been concluded that this technique for manoeuvring spacecraft into synchronous orbit is generally attractive and can offer economic benefits, if the correct approach is adopted in each case. If relatively small spacecraft are being considered, say of 1000 kg or less, in circular parking orbits of 300 to 5000km altitude, the use of dedicated ion thrusters for the
068 orbit transfer manoeuvre could be advocated.

The multi-kW solar arrays needed would be considered as part of the useful payload when the synchronous orbit was reached, because they would then be used to power the communications equipment. The payload ratio M_{pl}/M_o could then approach 70%, with transfer times of about 150 days. In addition, thrusters using T4A/T5 technology, with little extrapolation, could be used. Two or three such spacecraft could be launched on a single Ariane vehicle.

For larger spacecraft, the power levels required to enable geostationary orbit to be reached in a reasonable time appear to be in excess of those that can be usefully employed operationally. If this is the case, most of the mass of the solar arrays must be ascribed to the propulsion system, and M_{pl}/M_o is between 0.4 and 0.6, depending on transfer time. Although giving a larger payload in the final orbit than a chemical system, it is likely to be so costly to employ a dedicated electric propulsion system that it cannot be recommended for this purpose. However, the situation is considerably changed if a reusable space tug is considered. This would be equipped with solar arrays producing tens of kW and could be used to position many spacecraft in synchronous orbit before the arrays degrade to an unacceptable level. Such a tug vehicle has been studied extensively in both the USA and the UK, and it has been found economically viable. The same design could also be used with great advantage for many interplanetary missions. However, it must be concluded that the concept of a solar-electric tug is more compatible with the Space Shuttle than with Ariane, due to the need to refuel it at the end of each two-way mission. In addition, if used in conjunction with the Shuttle, it could return failed spacecraft from synchronous orbit for examination and repair.

LIST OF SYMBOLS

a_f	semi-major axis of final orbit
a_o	semi-major axis of initial orbit
a_t	semi-major axis at time t
e	electronic charge
F	total thrust applied to spacecraft
F_T	thrust of single ion thruster
h_o	altitude of initial circular orbit
M_f	mass of spacecraft in final orbit
M_o	mass of spacecraft in initial orbit
M_p	mass of propellant consumed during orbit transfer
M_{pl}	useful payload in final orbit
M_t	spacecraft mass at time t
m_i	ion mass
$\dot{m}_T$	total rate of propellant consumption of a thruster
n	number of thrusters used to provide total thrust F
P	total power required for orbit transfer manoeuvre with thrust F
P_a	power consumed by auxiliary thruster components
P_f	power available to spacecraft in synchronous orbit
P_i	input power to single ion thruster system
P_o	power available to spacecraft in initial orbit
P_T	power consumed by a thruster producing thrust F_T
P_t	power available to spacecraft at time t during orbit transfer
T	time taken for orbit transfer in absence of shadowing or air drag
T_s	time taken for orbit transfer, including shadowing and air drag
t	time after commencement of orbit transfer
$\overline{U}$	shadowing factor
V_T	net beam accelerating voltage of an ion thruster
v	velocity of singly charged beam ions
v_e	effective exhaust velocity of ion thruster system
ε	thruster discharge chamber efficiency in units of (eV/ion) $\times$ (e/m_i)
η_m	overall mass utilisation efficiency
η_p	electrical efficiency of power conditioner
μ	gravitational constant of the earth = $3.986 \times 10^{14}\ m^3/s^2$

REFERENCES

No.	Author	Title, etc
1	G.E. Keyte R.E.W. Jansson	The use of augmented electrothermal hydrazine thrusters in station-keeping applications. Proc Internat Conf on Properties of Hydrazine and its Applications as a Source of Energy, Poitiers, France (1974)
2	W.T. Lord R.C. Parkinson K. Allen J.A. Donovan P.J. Sherwood	Hydrogen resistojets for primary propulsion of communications satellites. Rocket Propulsion Establishment Technical Memorandum 605 (1972)
3	T. Rees D.G. Fearn	Use of the UK 10cm ion thruster for north-south station-keeping of the proposed European TVBS spacecraft. AIAA Paper 76-1059 (1976)
4	J.J. Pearson	Results of the definition study for the application of electric propulsion to the station-keeping of European Communication Satellites. Proc Conf on Electric Propulsion of Space Vehicles, Culham Lab, UK. Institution of Electrical Engineers Publication 100, pp 187-191 (1973)
5	Ford Aerospace	Intelsat V Spacecraft 1st Quarterly Report 21 September 1976 to 31 December 1976, Contract IS-796, Report WDL-TR7399-1.
6	D.G. King-Hele	The enlargement of elliptic satellite orbits by continuous micro-thrust. Proc 14th Int Astronaut Cong, Paris, Vol IV, pp 155-171 (1963), Springer-Verlag, Berlin (1965).
7	E.G.C. Burt	Space science and electrical propulsion. RAE Technical Report 68067 (1968)
8	E.G.C. Burt	On space manoeuvres with continuous thrust. RAE Technical Report 66149 (1966), Planetary and Space Sci, 15, 103-122 (1967)

06

REFERENCES (continued)

No.	Author	Title, etc
9	G.J. Davison	Orbit expansion by microthrust. Part 1. Description of results. RAE Technical Report 67249 (1967)
10	A.J. Sarnecki	Orbit expansion by microthrust. Part 2. Analysis. RAE Technical Report 68008 (1968)
11	P.D. Reader J.D. Redetz	A Delta-boosted, electrically-raised high power synchronous satellite. AIAA Paper 69-1104 (1969)
12	R. Ghosh G. Huson	Achievement of synchronous orbit using electric propulsion. AIAA Paper 69-275 (1969)
13	N.E. Ives	Power decay characteristics for a solar-electrically propelled spacecraft during orbit expansion to synchronous height. RAE Technical Report 71075 (1971)
14	W.T. Lord R.C. Parkinson	Two-stage (resistojet/ion motor) transfer of communications satellites. AIAA Paper 70-1116 (1970)
15	B.P. Day R. Hastings	The RAE 10cm hollow cathode mercury thruster. Paper 71-028, DGLR-Symposium Elektrische Antriebssysteme, Braunschweig (June 1971). DLR Mitteilung 71-21, pp 113-123 (October 1971), RAE Technical Report 71135 (1971)
16	B.P. Day F.C. Treble	The ion engine and large solar array for the X-5 spacecraft RAE Technical Report 68191 (1968)
17	Staff of Space Department, RAE	The UK ion thruster system and a proposed future programme. RAE Technical Report 77063 (1977)
18	S.P. Horio C.L. Watkins J.M. Shollenberger	Design synthesis of the SEPS. AIAA Paper 75-371 (1975)

068

REFERENCES (concluded)

No.	Author	Title, etc
19	R. Holdaway R.E.W. Jansson	The feasibility of geostationary emplacement of satellites by solar electric tug. J Brit Interplan Soc, 26, 220-227 (1973)
20	R.C. Parkinson	Cost optimisation of a resupplied communications satellite system with mixed-thruster electric propulsion. RPE Technical Report TR20 (1974)
21	B.P. Day M.D. Palmer	Ion thrusters and spacecraft configurations. Proc Conf on Electric Propulsion of Space Vehicles, Culham Laboratory (1973) IEE Conference Publication No. 100, pp 171-175 (1973)
22	ESA	ESA programme Ariane. ESA document IMP PDS 609.12.47 (September 1975)
23	D.G. Fearn R.C. Hughes	The T5 10cm mercury ion thruster system. AIAA Paper 78-650 (1978)
24	W.R. Kerslake R.G. Goldman W.C. Nieberding	SERT II: Mission, thruster performance, and in-flight thrust measurements. J Spacecraft and Rockets, 8, 213-224 (1971)
25	R.T. Bechtel V.K. Rawlin	Performance documentation of the engineering model 30 cm diameter thruster. AIAA Paper 76-1033 (1976)
26	J.E. Cake G.R. Sharp J.C. Oglebay F.J. Shaker R.J. Zavesky	Modular thrust sub-system approaches to solar electric propulsion module design. AIAA Paper 76-1062 (1976)
27	F.C. Treble	The RAE lightweight solar array. RAE Technical Report 73172 (1974)

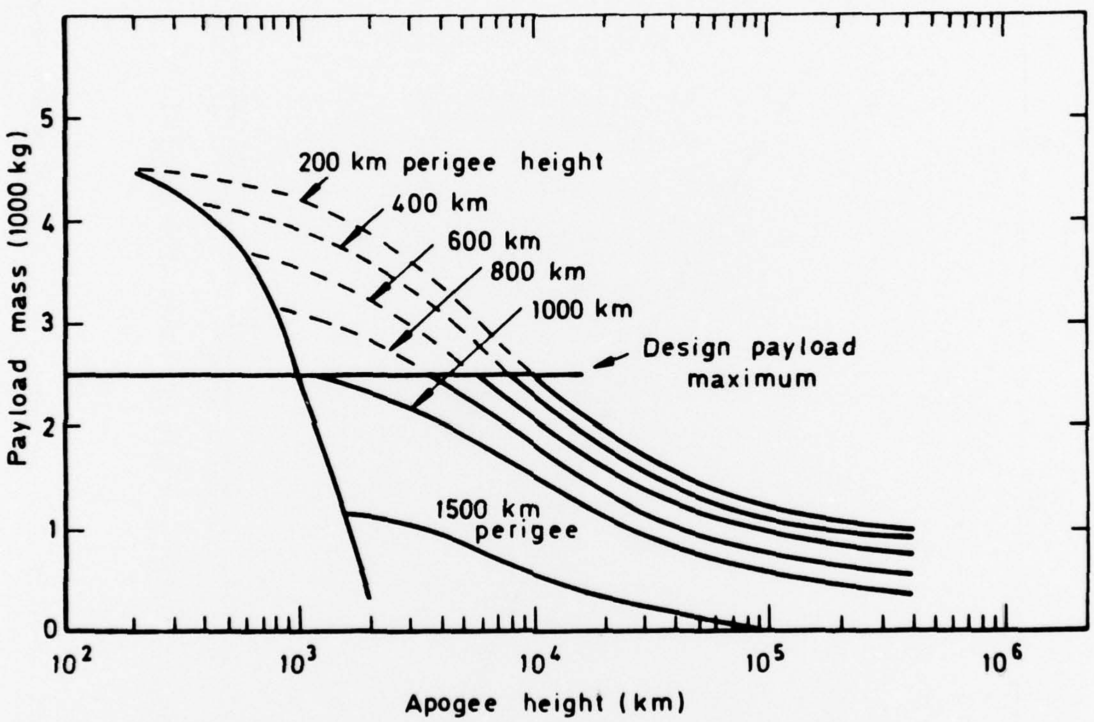

Fig 1 Performance of Ariane launcher, giving payload as a function of apogee and perigee heights (from ESA IMP PDS 609.12.47)

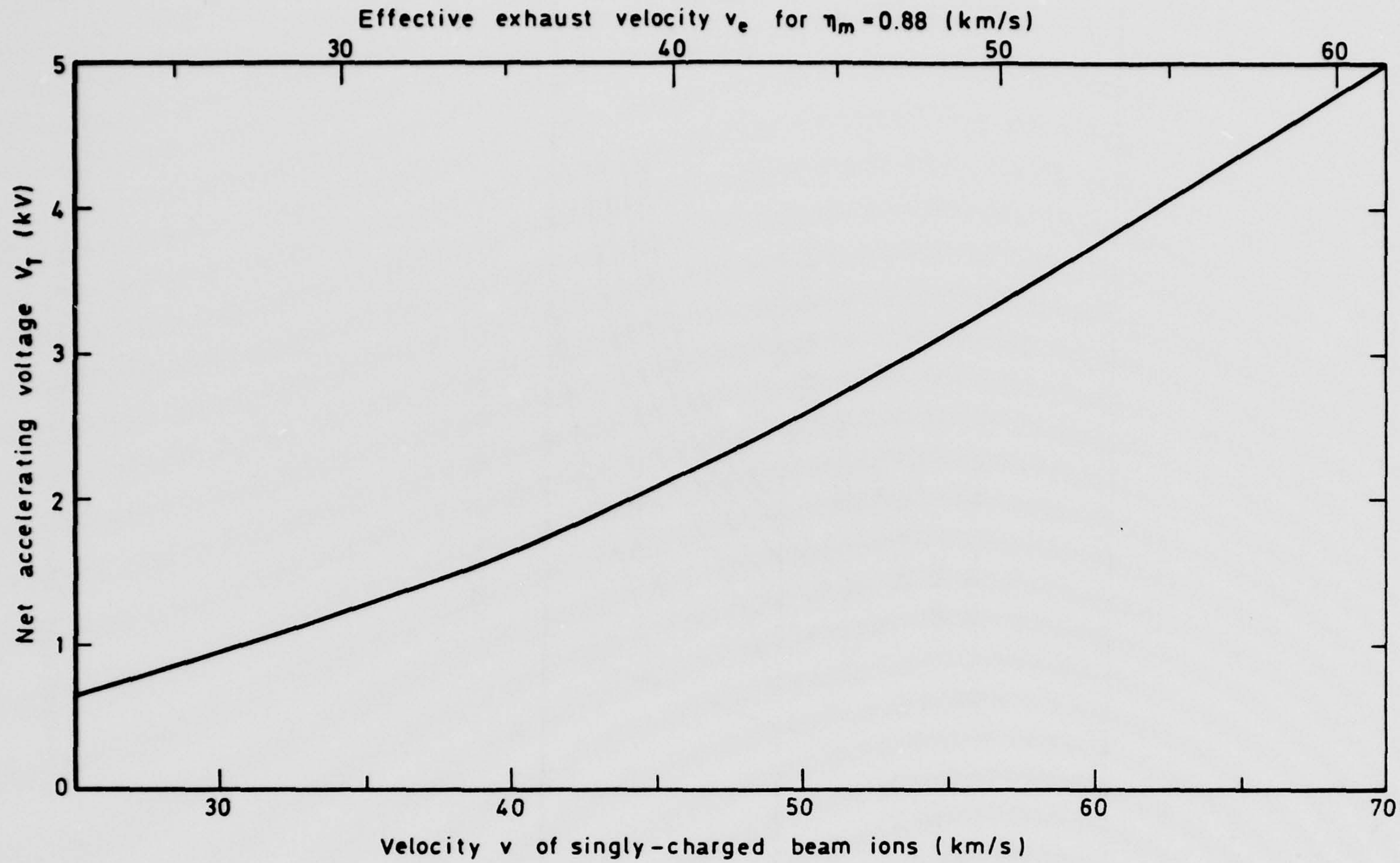

Fig 2 Ion beam velocity and effective exhaust velocity as functions of thruster potential

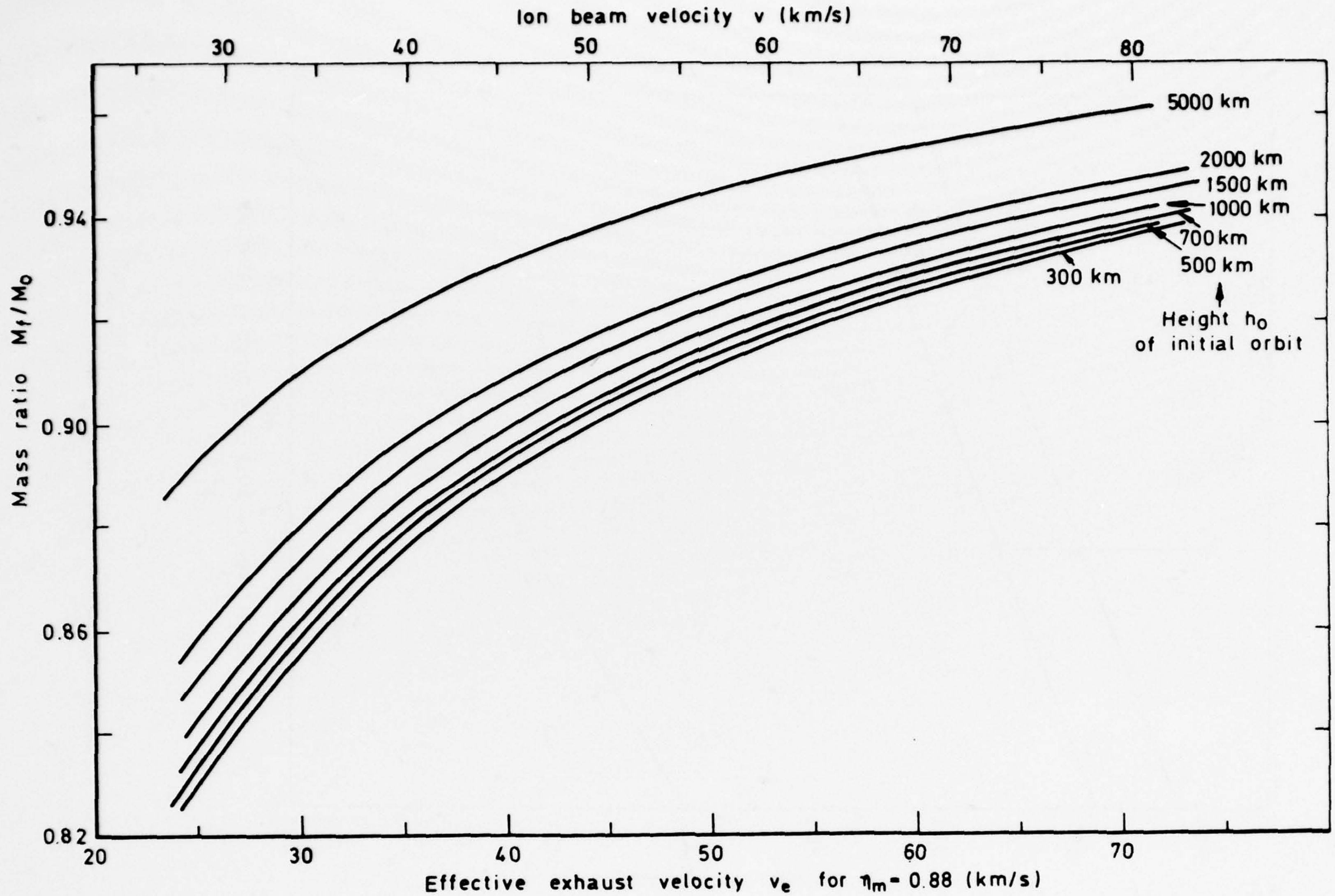

Fig 3 Mass ratio as a function of exhaust velocity and height of the initial circular orbit

Fig 4

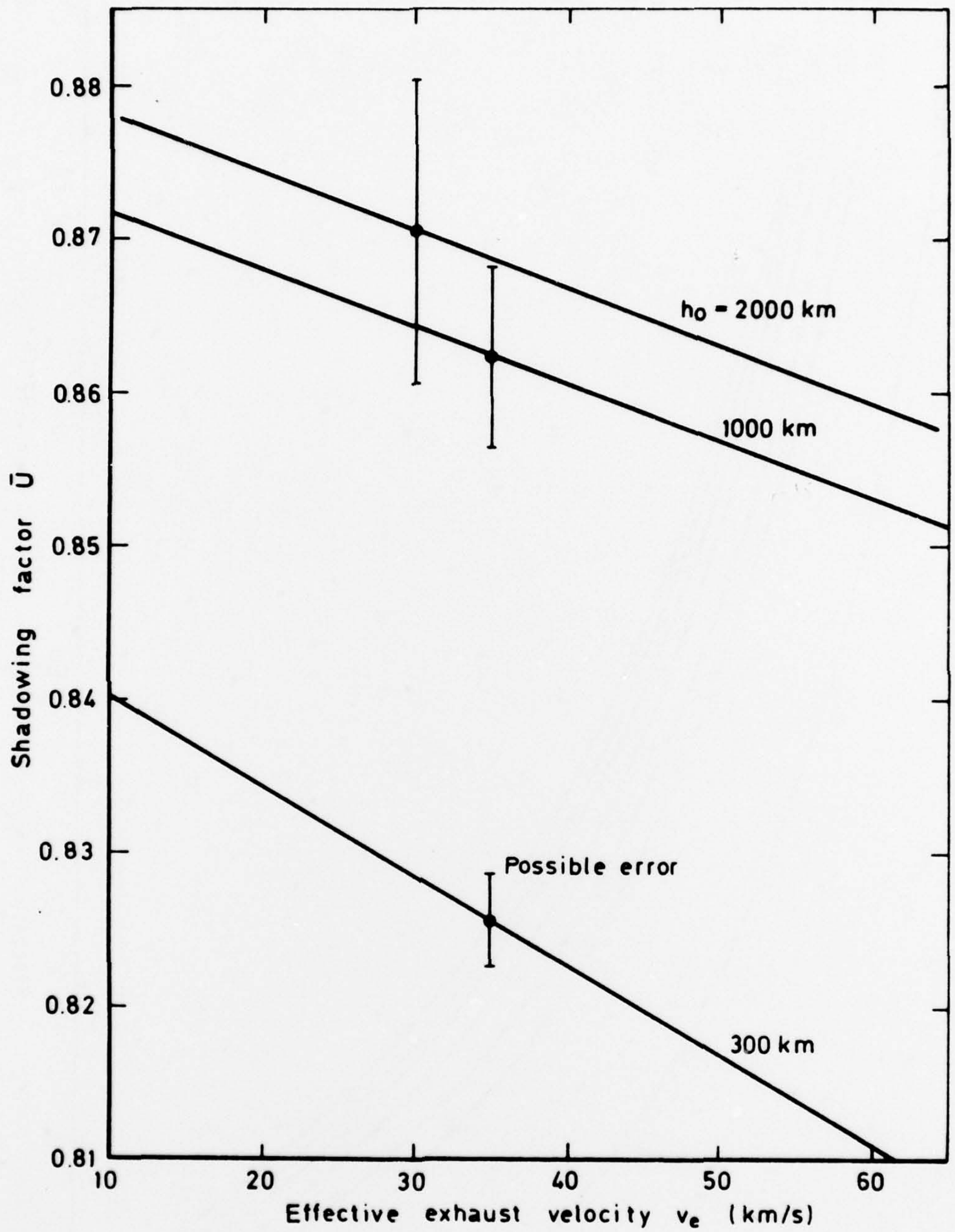

Fig 4 Shadowing factor as a function of v_e for circular initial orbits

TR 78068

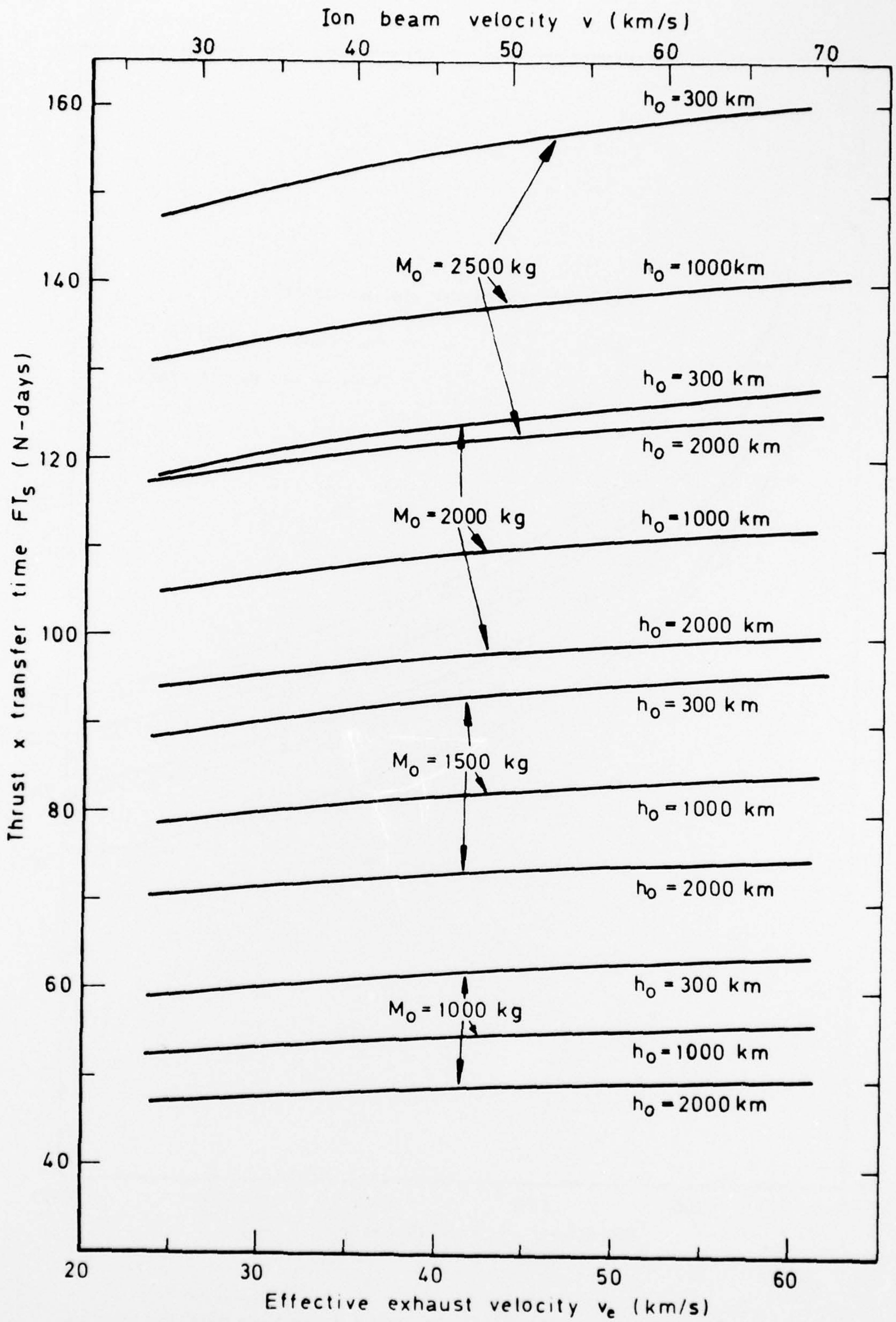

Fig 5 Thrust x transfer time as a function of v_e, initial mass and altitude of the initial circular orbit

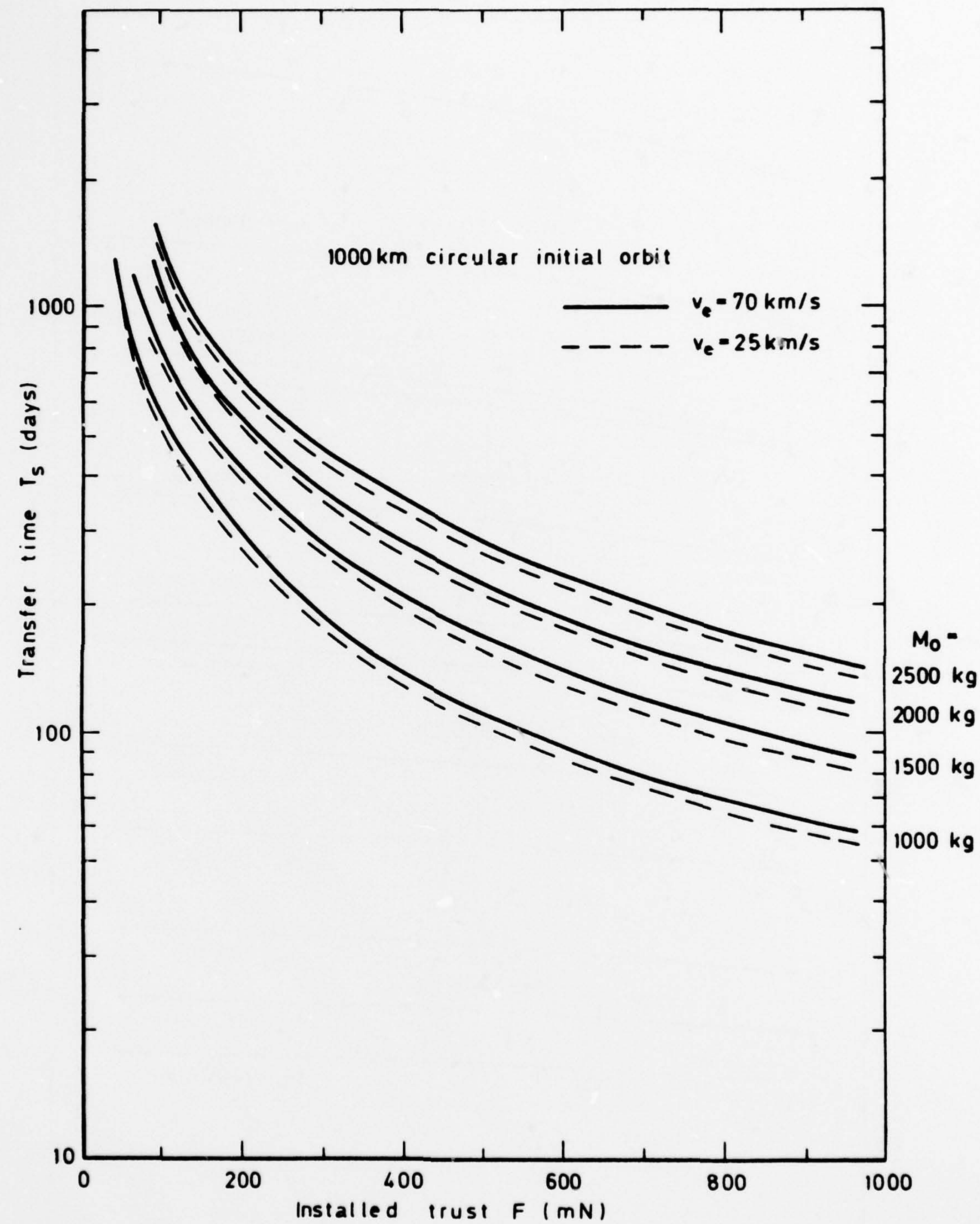

Fig 6 Transfer time as a function of installed thrust for a 1000 km circular initial orbit

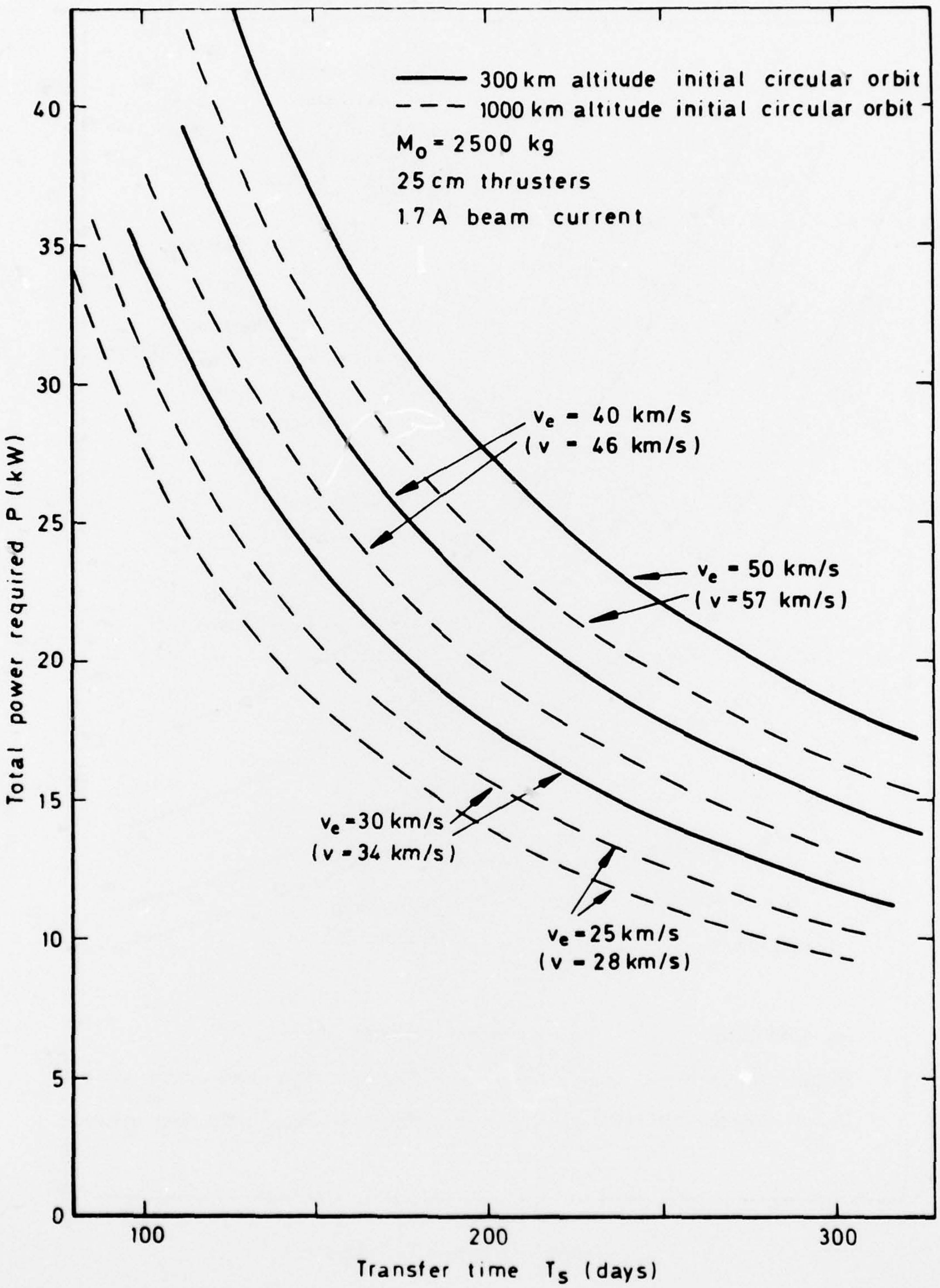

Fig 7 **Power required as a function of transfer time, exhaust velocity and initial orbit, for M_o = 2500 kg**

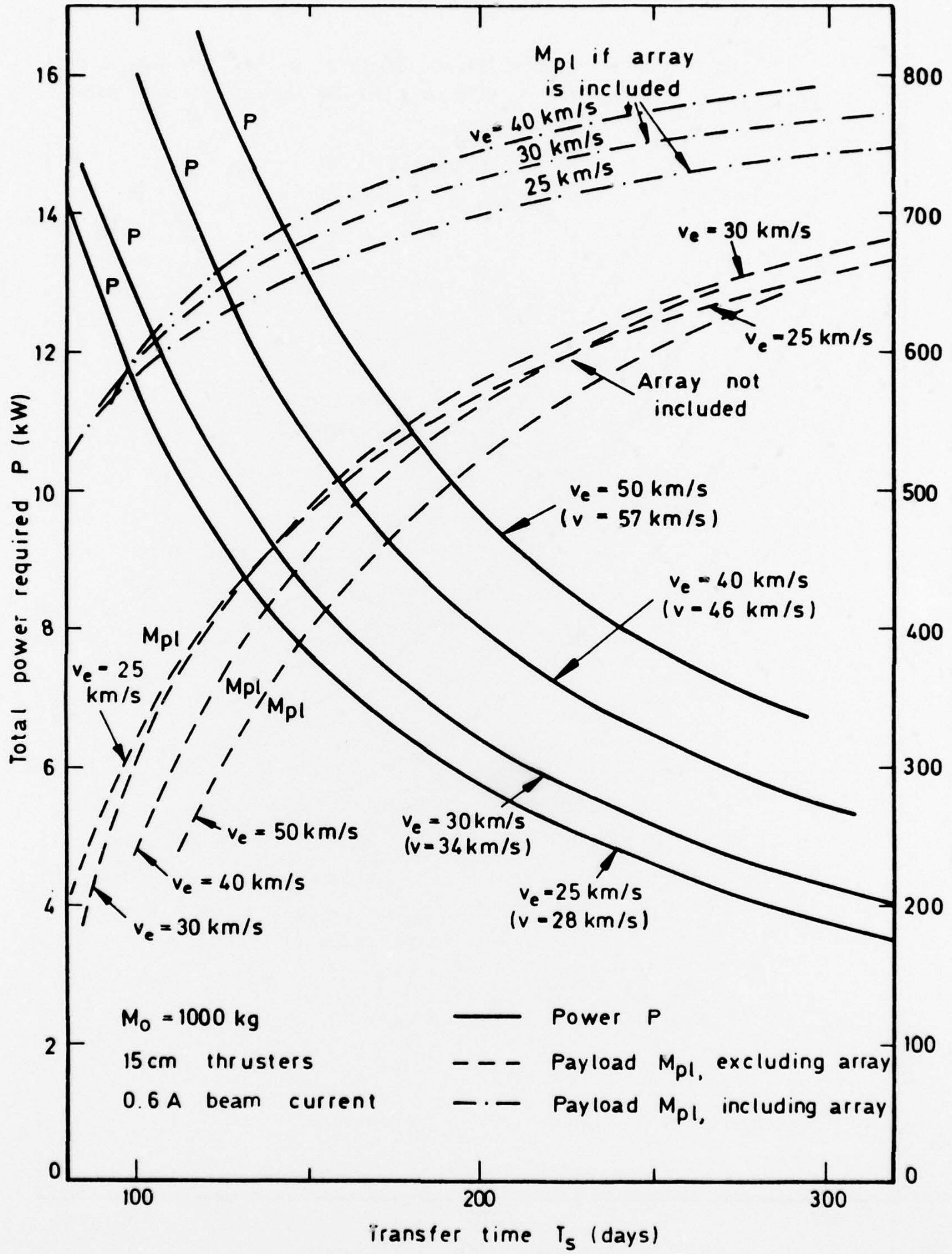

Fig 8 Power required and payload as functions of transfer time and exhaust velocity for M_o = 1000 kg and h_o = 1000 km

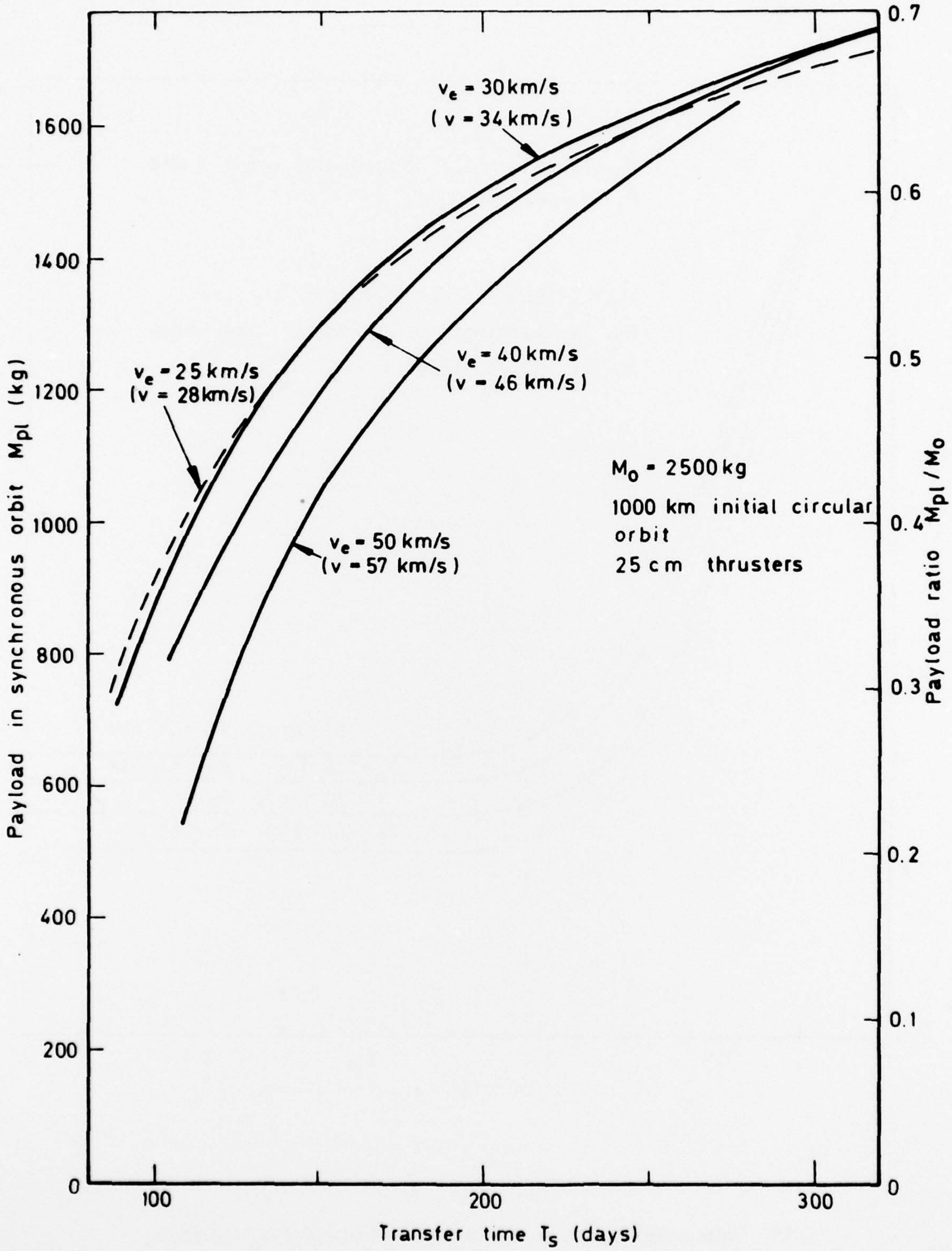

Fig 9 Payload and payload ratio as functions of transfer time and exhaust velocity for M_o = 2500 kg and h_o = 1000 km

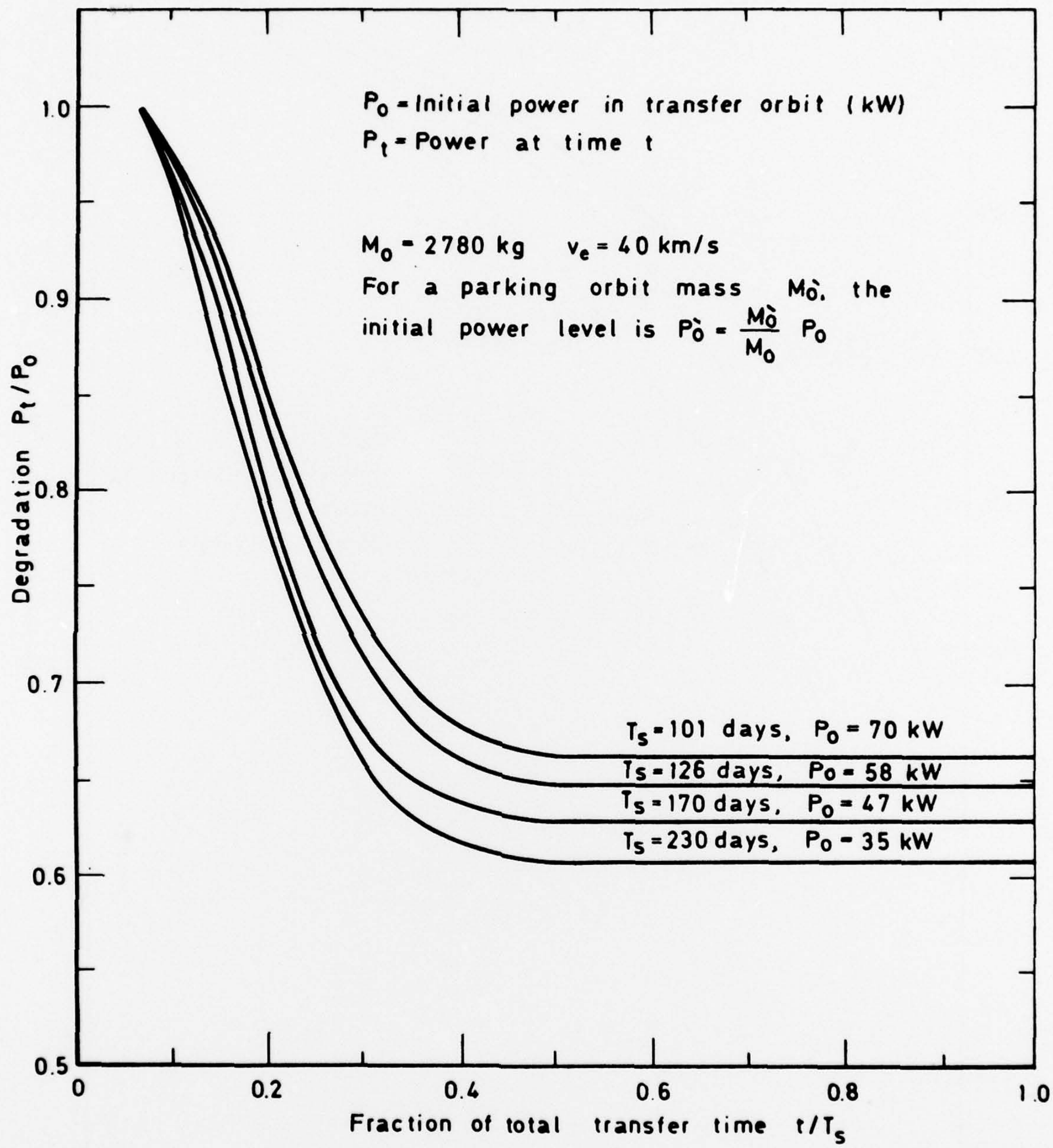

Fig 10 Solar array degradation as a function of time for M_0 = 2780 kg, h_0 = 500 km and v_e = 40 km/s

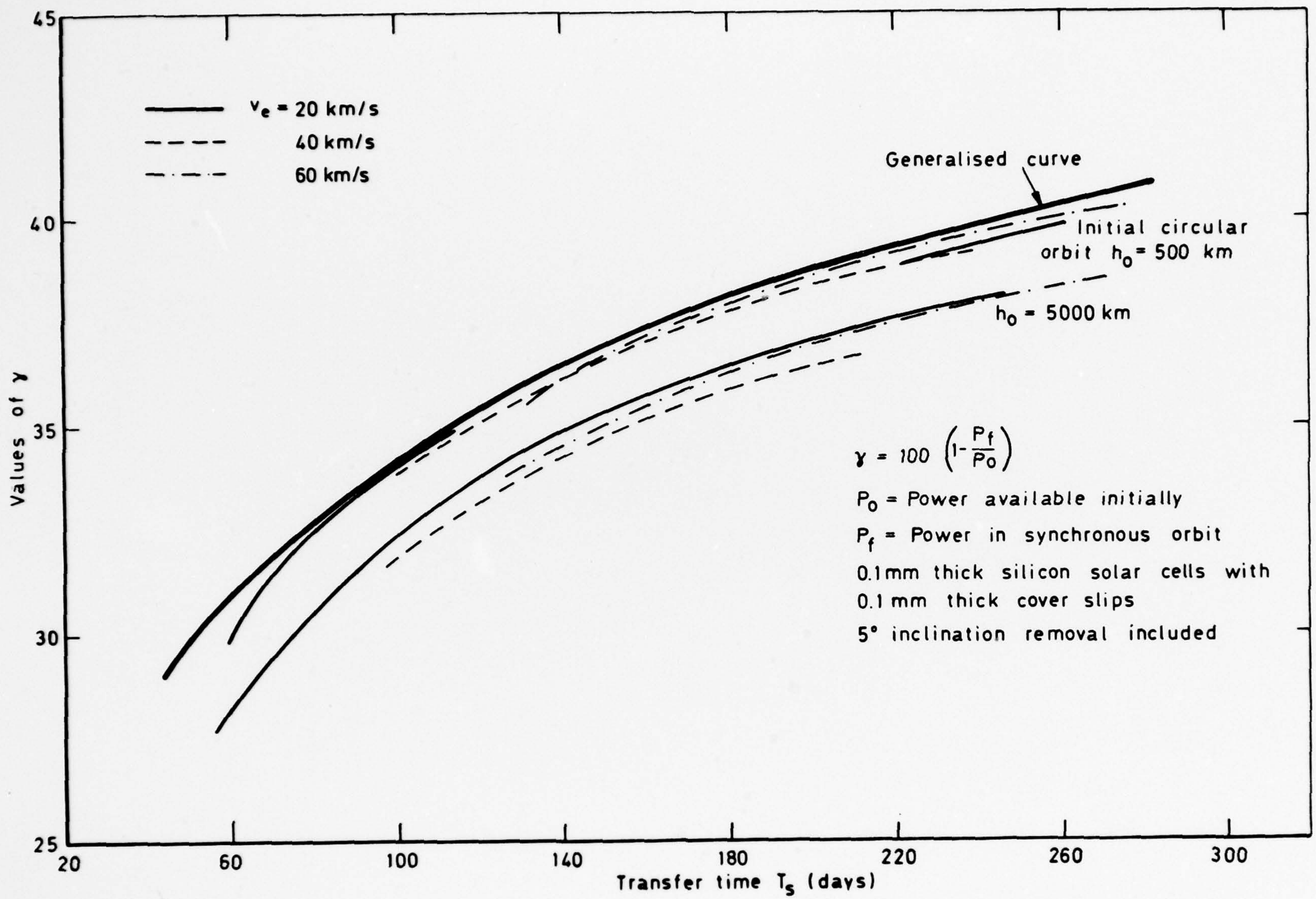

Fig 11 Percentage power loss as a function of transfer time, for two initial parking orbits

REPORT DOCUMENTATION PAGE

Overall security classification of this page

UNCLASSIFIED

As far as possible this page should contain only unclassified information. If it is necessary to enter classified information, the box above must be marked to indicate the classification, e.g. Restricted, Confidential or Secret.

1. DRIC Reference (to be added by DRIC)	2. Originator's Reference	3. Agency Reference	4. Report Security Classification/Marking
	RAE TR 78068	N/A	UNCLASSIFIED

5. DRIC Code for Originator	6. Originator (Corporate Author) Name and Location
850100	Royal Aircraft Establishment, Farnborough, Hants, UK
5a. Sponsoring Agency's Code	**6a. Sponsoring Agency (Contract Authority) Name and Location**
N/A	N/A

7. Title The use of ion thrusters for orbit raising

7a. (For Translations) Title in Foreign Language

7b. (For Conference Papers) Title, Place and Date of Conference

8. Author 1. Surname, Initials	9a. Author 2	9b. Authors 3, 4	10. Date	Pages	Refs.
Fearn, D.G.			June 1978	31	27

11. Contract Number	12. Period	13. Project	14. Other Reference Nos.
N/A	N/A		Space 551

15. Distribution statement
(a) Controlled by – MOD(PE) ASA6
(b) Special limitations (if any) –

16. Descriptors (Keywords) (Descriptors marked * are selected from TEST)

Electric propulsion. Orbit raising. Primary propulsion. Ion thrusters.

17. Abstract

Earlier analytical results for changing the orbital altitude of a spacecraft, using a tangentially thrusting electric propulsion system, have been employed to confirm that the mass of propellant required to attain a geostationary orbit is only a very small fraction of the initial mass of the satellite. However, such an orbit transfer technique requires a relatively long period of time compared with chemical propulsion, typically 100 to 300 days, and, for spacecraft masses of 1000 kg or more, multi-kW solar arrays are necessary. Unfortunately, the long transfer time also leads to a significant degradation of the power produced by these solar arrays, owing to the impact on them of energetic particles while traversing the earth's radiation belts. Nevertheless, it is concluded that this technique for manoeuvring a satellite into synchronous orbit can offer attractive economic benefits for two broad classes of spacecraft. These are relatively small satellites, of 1000 kg or less, for which a dedicated ion thruster system can be advocated, and much larger devices, for which a reusable solar electric tug vehicle having its own solar arrays would be appropriate. In the former case, a payload ratio of about 0.7 could be achieved, with a transfer time of 150 days, and two or three spacecraft could be launched on a single Ariane vehicle.

F5910/1